NASA TN D-1369

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TECHNICAL NOTE

D-1369

TEMPERATURE CONTROL OF THE EXPLORER IX SATELLITE

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON July 1962

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SUMMARY

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An investigation was made to determine a method of controlling the temperature of the transmitter and battery units of the radio tracking beacon on the inflatable Explorer IX (1961 Delta 1) satellite. The spherical satellite had a 12-foot diameter and was constructed of aluminum-foil—plastic-film laminate. It was determined that proper temperature control can be obtained by spotting the outside surface of the balloon with a coating having a low ratio of absorptance to emittance, for control while the satellite is in sunlight, and by physically separating the beacon transmitter and battery units from the inside surface of the balloon in order to minimize the rate of heat transfer for control while the satellite is in the earth's shadow. The analysis indicated that the inside surface of the balloon should have a reasonably high emittance; therefore, plastic film was utilized for the inside surface.

Successful transmission from the beacon during the sunlit phase of its first orbit indicated that the beacon set survived the launch, ejection, and inflation and that using the two hemispheres of the satellite itself as the antenna was successful. Subsequent failure of the beacon during its first orbit is attributed to excessive voltage being applied to the transmitter, with consequent destruction of the transistors.

INTRODUCTION

The present values for the atmospheric density at satellite altitudes are inferred from the orbital decay rates of satellites of various sizes and shapes (refs. 1, 2, and 3). Many of these satellites are not suited for this purpose because their size and weight are such that their orbital decay rates are inconveniently slow, and also because their frontal areas are not constant. Since orbital decay rates are more rapid for satellites having a low ratio of mass to area, a lightweight inflatable spherical satellite having a 12-foot diameter was designed and constructed by the Langley Research Center (ref. 4). It was put into orbit on February 16, 1961 by the Scout launching vehicle and designated Explorer IX (1961 Delta 1). The successful launching followed a series of unsuccessful attempts with modified Redstone and Juno vehicles. In these earlier satellite launching attempts, it was planned to rely on worldwide optical tracking of the satellite to obtain the drag data, with a limited backup provided by long-range radar. With completion of the Minitrack network an additional method became available for tracking the satellite. If the Minitrack network were used in addition to optical tracking, the satellite could be tracked day and night regardless of weather conditions and thus the quantity of data could be increased. For this reason, it was decided to place a radio tracking beacon set on the satellite for launchings attempted by the Scout vehicle. The radio beacon set would back up the worldwide optical tracking system. The addition of a radio tracking beacon required special electronic packaging techniques along with thermal design changes to the satellite because, although the sphere itself could stand temperatures up to 175° C, the lifetimes of the transistors and storage battery cells of the radio beacon set are greatly reduced for temperatures below -10° C or above 60° C.

The purpose of the present paper is to describe the methods used to provide the desired temperature control for the beacon transmitter set while the satellite is in sunlight or in the earth's shadow and to present an evaluation of this temperature control as concluded from the observed mode of operation in orbit. Much of the investigation was carried out in conjunction with the Astro-Electronics Division of Radio Corporation of America, contractor for the radio tracking beacon transmitter set.

SYMBOLS

In case it is desired to convert distances given in the English system of units to metric units, the following relationships apply: 1 foot = 0.3048 meter, and 1 statute mile = 5,280 feet = 1609.344 meters.

A	area, cm^2
a	albedo of earth, 0.36
c_p	specific heat, cal/gram-°C
C_S	solar constant, 2.00 cal/cm^2-min
D	diameter of inflatable satellite, cm unless otherwise noted
h	satellite altitude, statute miles

$K = \dfrac{r_E}{r_E + h}$

M mass of satellite, grams

m mass of transmitter unit, grams

q rate of transfer of thermal energy, cal/cm^2-min

r_E radius of earth

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T temperature, $^{\circ}K$

$\overline{T^4}$ fourth power of skin temperature averaged over surface, $^{\circ}K^4$

t time, min

α absorptance

ϵ emittance

ρ density, $grams/cm^3$

σ Stefan-Boltzmann constant, 8.135×10^{-11} cal/cm^2-min-$^{\circ}K^4$

Subscripts:

A aluminum foil

B battery or transmitter unit

C coldest area

d transmitter-unit surface facing main internal volume of satellite

E earth, earth's radiation

g thermal-gap surfaces

H hottest area

i inside surface of balloon satellite

my plastic film (Mylar)

o outside surface of balloon satellite

S solar, solar radiation

s satellite skin

t total surface

w white-epoxy-enamel surface

1 internal dissipation for transmitter or battery unit

2 balloon internal radiation while satellite is in sunlight L1952

3 balloon internal radiation while satellite is in the earth's shadow

DESCRIPTION OF SATELLITE

The Explorer IX satellite is an inflatable balloon structure having a 12-foot diameter. It is constructed of a four-ply laminate consisting of alternate layers of 0.5-mil-thick aluminum foil and 0.5-mil-thick plastic film (Mylar). One layer of the aluminum foil forms the outside surface of the satellite and one layer of the plastic film forms the inside surface. The radio tracking beacon transmitter was powered by four groups of solar cells and a storage-battery power supply. Shown in figure 1 is the wiring diagram showing how all components of the radio tracking beacon transmitter set were connected. The satellite is constructed of two hemispheres connected by an equatorial insulating band of $1\frac{1}{2}$- inch-wide Mylar. The radio-frequency output of the beacon transmitter was connected to the two hemispheres and thus the balloon satellite itself was utilized as the transmitting antenna.

The solar cells are located on the outside of the satellite skin at the apexes of an imaginary regular tetrahedron in order to provide continuous charging of the storage battery cells while the satellite is in sunlight, regardless of satellite orientation. The cells of the storage-battery power supply and transmitter are distributed into two units. One unit contains the transmitter and four battery cells, and the other unit contains six battery cells. The entire satellite with its radio tracking beacon transmitter set weighs only 14.62 pounds so it has a very low mass per unit area. The satellite has sufficient rigidity to remain spherical without internal pressure after the inflation gas has leaked out and it is a good reflector of sunlight for optical tracking. Figure 2 is a cutaway drawing of the 12-foot-diameter satellite as it would appear in orbit. The figure shows the location of the tracking beacon transmitter unit and the battery unit on the inside surface of the satellite skin. The insulating band and one of the groups of solar cells are also shown along with the printed cable used to electrically connect the components of the radio tracking beacon transmitter set.

Figure 3 is a photograph of the 12-foot-diameter inflatable satellite with the thermal coating of white epoxy enamel. The white epoxy enamel was applied in the form of dots mostly 2 inches in diameter, distributed evenly over the satellite skin surface to give proper temperature control regardless of satellite orientation in sunlight. In the vicinity of the transmitter and battery units, the diameter of the dots was reduced to 1 inch to obtain a more uniform temperature distribution. The satellite skin directly over the transmitter and battery units was coated with a 7- by 9-inch area of white epoxy enamel to reduce the effect of direct solar radiation on the units. The plastic insulating band was coated with white enamel so that sunlight would not be transmitted through it and fall upon the two units.

THERMAL CALCULATIONS AND DESIGN

Satellite Skin Temperatures

<u>Derivation of equations for maximum and minimum equilibrium temperatures in sunlight and in the shadow of the earth</u>.- Since the satellite was to be at altitudes such that aerodynamic heating would be negligible, it would receive heat by radiation only. Radiation absorbed by the satellite while in sunlight consists of three parts (ref. 5): Direct radiation from the sun, $\frac{\pi D^2}{4} C_S \alpha_S$; solar radiation reflected from the earth, the maximum being $\frac{\pi D^2}{4} C_S \alpha_S a \; 2\left[1 - \left(1 - K^2\right)^{1/2}\right]$; and direct radiation from the earth, $\frac{\pi D^2}{4} C_S \frac{1 - a}{4} \alpha_E \; 2\left[1 - \left(1 - K^2\right)^{1/2}\right]$. The satellite receives only direct earth radiation while it is in the shadow of the earth. By equating the radiant energy reception for the satellite and the radiation energy loss, expressions can be obtained for the hottest and coldest skin equilibrium temperatures in sunlight and in the earth's shadow. In sunlight the highest temperature of the satellite skin is given by the equation

$$(\epsilon_i + \epsilon_o)\sigma T_H^{\,4} = C_S\alpha_S + \frac{1}{4}\frac{\epsilon_i}{\epsilon_o} C_S\alpha_S \left\{1 + 2\left(a + \frac{1 - a}{4}\frac{\alpha_E}{\alpha_S}\right)\left[1 - \left(1 - K^2\right)^{1/2}\right]\right\} \tag{1}$$

The derivation of this equation can be found in reference 5. In a similar manner an expression for the coldest equilibrium temperature of the satellite skin when the satellite is directly between the sun and the earth can be derived. The rate of radiation from both sides of a unit area of balloon skin is $(\epsilon_i + \epsilon_o)\sigma T_C^{\,4}$. The rate of reception of energy by unit area of the coldest spot would be no less than that from internal radiation exchange only. This rate is given in reference 5 and is $\epsilon_i\sigma\overline{T^4}$ or

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$$\frac{1}{4}\frac{\epsilon_i}{\epsilon_o} C_S\alpha_S \left\{1 + 2\left(a + \frac{1 - a}{4}\frac{\alpha_E}{\alpha_S}\right)\left[1 - \left(1 - K^2\right)^{1/2}\right]\right\} \tag{2}$$

Equation (2) applies only when the satellite is located directly between the sun and the earth. Then, in equilibrium, by equating the reception and radiation loss, the expression for the coldest equilibrium temperature of the satellite skin when the satellite is directly between the sun and the earth is

$$(\epsilon_i + \epsilon_o)\sigma T_C^{\,4} = \frac{1}{4}\frac{\epsilon_i}{\epsilon_o} C_S\alpha_S \left\{1 + 2\left(a + \frac{1 - a}{4}\frac{\alpha_E}{\alpha_S}\right)\left[1 - \left(1 - K^2\right)^{1/2}\right]\right\} \tag{3}$$

In the shadow of the earth the lowest temperature of the satellite skin is given by the equation

$$(\epsilon_i + \epsilon_o)\sigma T_C^{\,4} = \frac{1}{4}\frac{\epsilon_i}{\epsilon_o} C_S \frac{1 - a}{4} \alpha_E\, 2\left[1 - \left(1 - K^2\right)^{1/2}\right] \tag{4}$$

The derivation of this equation is found in reference 5. The derivation of the expression for the hottest equilibrium temperature of the satellite skin while in the shadow of the earth is similar to that of the maximum temperature in sunlight. For unit area of the hottest spot facing the earth, the thermal balance is given by the equation

Reception from earth + Reception from inside of satellite

= Radiation from both sides of satellite skin

If the unit-area element facing the earth is considered as a flat plate then its rate of reception of radiation from the earth is $C_S \frac{1 - a}{4} \alpha_E K^2$. (See ref. 6.) The rate of reception from the inside of the satellite is (ref. 5)

$$\epsilon_i \sigma \overline{T^4} = \frac{1}{4} \frac{\epsilon_i}{\epsilon_o} C_S \frac{1 - a}{4} \alpha_E \, 2\left[1 - \left(1 - K^2\right)^{1/2}\right] \tag{5}$$

The rate of radiation from both sides of the satellite skin is $(\epsilon_i + \epsilon_o)\sigma T_H^{\,4}$. Therefore, in thermal equilibrium, the equation for maximum temperature of the satellite skin in the shadow of the earth is

$$(\epsilon_i + \epsilon_o)\sigma T_H^{\,4} = C_S \frac{1 - a}{4} \alpha_E \left\{\frac{1}{2} \frac{\epsilon_i}{\epsilon_o}\left[1 - \left(1 - K^2\right)^{1/2}\right] + K^2\right\} \tag{6}$$

<u>Effect of emittance of satellite skin inner surface</u>.- The early 12-foot-diameter inflatable spheres were constructed of a three-layer laminate with aluminum foil on both sides of the plastic film. The addition of the radio tracking beacon to the sphere surface made it necessary to change from the three-layer laminate to a four-layer laminate having aluminum foil for the outside surface and plastic film for the inside surface. The advantage is apparent from figure 4 which shows the maximum and minimum equilibrium skin temperatures in sunlight and in shadow for both laminate skin constructions plotted against satellite altitude. The temperatures in sunlight for the figure are those that occur when the satellite is directly between the sun and the earth. The experimental values of absorptance and emittance that were used in all calculations are shown in table I. Measurements of the absorptance and emittance of the aluminum-foil—plastic-film laminated balloon material and of a white epoxy enamel coating were made by the contractor for the radio tracking beacon transmitter set. Figure 4 shows that in sunlight the sphere with plastic film inside and aluminum foil outside has a lower maximum equilibrium temperature as well as a higher minimum equilibrium temperature than has the sphere with aluminum foil on both the inside and outside surfaces. The reason is that the higher emittance of the plastic film inside the sphere results in more internal radiation exchange between the high-temperature region and the low-temperature region. The same is also shown for the satellite in the earth's shadow.

<u>Effect of surface coating</u>.- Since the temperatures of the transmitter and battery units (which must remain between 60° C and -10° C for efficient operation) would be strongly influenced by the satellite skin temperature, the equilibrium temperatures shown in figure 4 would

be too high in sunlight and too low in the earth's shadow, even when the inside surface of the satellite is plastic film. Equation (1) indicates that the maximum equilibrium temperature of the satellite's skin in sunlight is principally dependent upon the ratio of absorptance to emittance. If the values of satellite solar absorptance and outside surface emittance can be adjusted by coating part or all of the satellite's outside surface area, then a means of controlling the satellite equilibrium temperature in sunlight is possible. Therefore, the satellite skin equilibrium temperatures in sunlight were reduced by applying to the outside surface a coating having a low ratio of solar absorptance to emittance.

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The 12-foot-diameter inflatable sphere was a fully designed and developed satellite when the radio tracking beacon transmitter set was placed on it. Any change in the surface of the sphere for temperature control had to be compatible with the design. Because the satellite thermal design had to be completed in a short period of time and the satellite was to have high reflectivity in both the radio and optical ranges of the spectrum, completely covering the outside surface was considered undesirable. The outside surface was covered over a small percentage with a coating having a low ratio of absorptance to emittance so that the reflectance for both light and radio waves would not be appreciably altered. The coating was applied as white dots distributed evenly over the surface. The equations for the effective absorptance and emittance would then be

$$\alpha_S = \alpha_{S,A}\left(1 - \frac{A_w}{A_t}\right) + \alpha_{S,w}\frac{A_w}{A_t} \tag{7}$$

and

$$\epsilon_o = \alpha_E = \epsilon_{o,A}\left(1 - \frac{A_w}{A_t}\right) + \epsilon_{o,w}\frac{A_w}{A_t} \tag{8}$$

White epoxy enamel was selected for the thermal-control coating because the enamel had a low ratio of absorptance to emittance, adhered satisfactorily to the aluminum surface, and was not affected appreciably by solar ultraviolet radiation. Presented in figure 5 are the maximum and minimum equilibrium skin temperatures in sunlight and the average equilibrium skin temperatures in the earth's shadow, for the 12-foot satellite with 10, 15, 20, and 25 percent of the outside surface area coated with white epoxy enamel, plotted against satellite altitude. The maximum and minimum equilibrium temperatures in sunlight are seen to decrease as the percentage of area covered by white epoxy enamel is increased. Figure 5 indicates that between 15 and 20 percent could possibly satisfy the beacon temperature requirements in sunlight, but in the earth's shadow, the equilibrium skin temperature would still be far below the -10° C specified for efficient beacon transmitter operation. Only one curve is shown for the average equilibrium skin temperature in the earth's shadow since there is no variation with outer surface absorptance and emittance. This is because the satellite is absorbing and emitting energy of the same spectral distribution, or $\epsilon_o = \alpha_E$. The thermal-balance equation for the average skin temperature of the satellite in the earth's shadow is given in reference 5 and is

$$\epsilon_o \sigma \overline{T^4} = \frac{1}{4} C_S \frac{1 - a}{4} \alpha_E 2\left[1 - \left(1 - K^2\right)^{1/2}\right] \tag{9}$$

Since $\epsilon_o = \alpha_E$, equation (9) reduces to

$$\sigma \overline{T^4} = \frac{1}{4} C_S \frac{1 - a}{2}\left[1 - \left(1 - K^2\right)^{1/2}\right] \tag{10}$$

which shows that the average skin temperature in the earth's shadow is a function of only geometry and the solar constant. Since the average temperature of the satellite when it is in the earth's shadow is not dependent on emittance and is well below an acceptable value of -10° C as is shown in figure 5, some other means of temperature control for the radio beacon set must be developed when the satellite is in the earth's shadow.

Thermal response.- The temperatures of figure 5 would be acquired if the satellite were to remain either in sunlight or in the earth's shadow for a sufficient length of time to reach thermal equilibrium. In order to determine whether the finite thermal capacity of the satellite might substantially reduce the range of temperature fluctuations, an approximate analysis (appendix) was made of the thermal response of the satellite throughout its entire orbit. The results are presented in figure 6 for circular orbits in the ecliptic having altitudes of 500 and 1,600 statute miles, with 17 percent of the outside surface area coated with white epoxy enamel. Figure 6 indicates that thermal equilibrium is practically attained in a relatively short time compared to the orbital period; that is, the temperature lag after the satellite enters the earth's shadow cannot suffice to keep the transmitter and battery units above -10° C if the units are attached directly to the skin of the sphere. In order to provide a sufficient thermal lag, the transmitter and battery units were physically separated from the satellite skin and provided with reflecting surfaces. The main features of the design are included here, but a description of the final engineering design of the radio beacon set is given in reference 7. Calculations to determine whether the thermal decoupling was adequate are presented in the following section.

Transmitter-Unit Temperatures

Maximum and minimum equilibrium temperatures in sunlight.- The maximum and minimum equilibrium temperatures for the transmitter and battery units while the satellite is in sunlight were obtained by thermal-balance calculations for the units. The thermal balance of the unit containing six storage battery cells was similar to that of the transmitter unit since the mass, volume, and general arrangement of both units were similar; therefore, only the thermal analysis for the transmitter unit will be discussed. Figure 7 shows the general arrangement of the transmitter unit. The unit consists of four button-type nickel cadmium storage battery cells and transmitter mounted on a fiber-glass circuit board with the assembly molded into polyurethane foam. Polished aluminum forms the outside cover of the transmitter unit and the reflecting gap surfaces adjacent to the satellite skin. Figure 8 is a schematic diagram of the transmitter unit and the satellite skin, showing the different components of the unit's thermal balance. The radiative thermal gap built into the transmitter unit was placed adjacent to the satellite skin and heat exchange between the satellite skin and the adjacent surface of the transmitter unit was considered as parallel-plate heat exchange. The heat exchange between the transmitter unit and the main internal surface of the satellite was considered as that between a small body and a large enclosing surface. The methods for treating these types of radiation heat exchange are discussed in reference 8. With these considerations the heat-balance equations for the transmitter unit in thermal equilibrium while the satellite is in sunlight were developed.

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The heat balance for the satellite skin adjacent to the transmitter unit is

$$C_S \alpha_{S,w} + \sigma\left(\frac{1}{\frac{1}{\epsilon_g} + \frac{1}{\epsilon_g} - 1}\right)\left(T_{B,H}^4 - T_{s,w}^4\right) = \epsilon_{o,w}\sigma T_{s,w}^4 \qquad (11)$$

where $\frac{1}{\frac{1}{\epsilon_g} + \frac{1}{\epsilon_g} - 1}$ is the radiative transfer factor for parallel plates and where the outside surface of the satellite skin in the immediate neighborhood of the transmitter unit is coated solid white to reduce the effect of direct solar radiation on the unit.

The heat balance for the transmitter unit is

$$q_2 + q_1 = \epsilon_d \sigma T_{B,H}^4 + \sigma\left(\frac{1}{\frac{1}{\epsilon_g} + \frac{1}{\epsilon_g} - 1}\right)\left(T_{B,H}^4 - T_{S,w}^4\right) \tag{12}$$

where

$$q_2 = \frac{1}{4}\frac{\epsilon_d}{\epsilon_o} C_S \alpha_S \left\{1 + 2\left(a + \frac{1 - a}{4}\frac{\alpha_E}{\alpha_S}\right)\left[1 - \left(1 - K^2\right)^{1/2}\right]\right\}$$

which is the rate radiation within the main volume of the satellite is absorbed per square centimeter, and where q_1 is the rate heat is dissipated internally in the transmitter unit based on maximum cross-sectional area. This internal dissipation is 0.05092 cal/cm^2-min.

Equations (11) and (12) were solved for T_S, the temperature of the satellite skin adjacent to the transmitter unit, and the two expressions were set equal to each other. By means of equations (7) and (8), α_S, ϵ_o, and α_E were replaced by the single variable A_w/A_t; thus, the following equation is given for the fraction of the skin area to be coated as a function of the maximum allowable temperature of the transmitter unit:

$$\frac{A_w}{A_t} = \frac{\left\{\left\{\frac{\epsilon_g\left[\sigma T_{B,H}^4\left(\epsilon_{o,w} + \epsilon_g\right) - C_S\alpha_{S,w}\right] + \epsilon_{o,w}\epsilon_d\left(2 - \epsilon_g\right)\sigma T_{B,H}^4}{\epsilon_g + \epsilon_{o,w}\left(2 - \epsilon_g\right)} - \frac{1}{4}\epsilon_d C_S \frac{1 - a}{2}\left[1 - \left(1 - K^2\right)^{1/2}\right] - q_1\right\}\epsilon_{o,A} - \frac{1}{4}\epsilon_d C_S \alpha_{S,A}\left\{1 + 2a\left[1 - \left(1 - K^2\right)^{1/2}\right]\right\}\right\}}{\left\{\frac{1}{4}\epsilon_d C_S\left(\alpha_{S,w} - \alpha_{S,A}\right)\left\{1 + 2a\left[1 - \left(1 - K^2\right)^{1/2}\right]\right\} + \left(\epsilon_{o,A} - \epsilon_{o,w}\right)\left\{\frac{\epsilon_g\left[\sigma T_{B,H}^4\left(\epsilon_{o,w} + \epsilon_g\right) - C_S\alpha_{S,w}\right] + \epsilon_{o,w}\epsilon_d\left(2 - \epsilon_g\right)\sigma T_{B,H}^4}{\epsilon_g + \epsilon_{o,w}\left(2 - \epsilon_g\right)} - \frac{1}{4}\epsilon_d C_S \frac{1 - a}{2}\left[1 - \left(1 - K^2\right)^{1/2}\right] - q_1\right\}\right\}} \tag{13}$$

A design value for the maximum equilibrium temperature of the transmitter unit while the satellite is in sunlight was selected as 55° C. For this maximum temperature at the expected perigee altitude of approximately 400 statute miles, equation (13) indicated that over 50 percent of the satellite surface must be coated with white epoxy enamel to obtain the desired temperature control in sunlight. This percentage was considered to be unreasonable. Preliminary analysis indicated that an increase in the emittance of the transmitter-unit surface facing the main internal volume of the satellite would result in a smaller required percentage of surface area coating for the balloon; however, an increase in this emittance would also result in a more rapid cooling rate for the transmitter unit when the satellite is in the earth's shadow. An increase in the emittance of the transmitter-unit surface couples it closer to the balloon temperature and would result in exceedingly low temperatures of the transmitter unit while the satellite is in the shadow of the earth. A compromise was made between the cooling rate of the transmitter unit and the required percentage of surface area coated. The emittance of the transmitter-unit surface facing the main internal volume of the satellite was increased to 0.15 by distributing black dots over the surface.

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With $\epsilon_d = 0.15$ and a transmitter-unit maximum equilibrium temperature of 55° C at a perigee of 400 statute miles, equation (13) indicated that 17 percent of the satellite surface area must be coated with white epoxy enamel to obtain the temperature control. Equations (11) and (12) were then solved for the transmitter-unit maximum equilibrium temperature in sunlight as a function of satellite altitude and the fraction of surface area coated with white epoxy enamel. This expression was

$$T_{B,H}^4 = \frac{\left[\epsilon_g + \epsilon_{o,w}\left(2 - \epsilon_g\right)\right]\left(q_1 + q_2\right) + C_S\alpha_{S,w}\epsilon_g}{\sigma\left[\epsilon_g\left(\epsilon_{o,w} + \epsilon_d - \epsilon_{o,w}\epsilon_d\right) + 2\epsilon_{o,w}\epsilon_d\right]} \tag{14}$$

where the satellite altitude and ϵ_o, α_E, and α_S are in the term q_2 and ϵ_o, α_E, and α_S are a function of A_w/A_t as in equations (7) and (8). By making the term, $C_S\alpha_{S,w}\epsilon_g$, equal to zero in equation (14), the minimum equilibrium temperature of the transmitter unit while the satellite is in sunlight can be calculated for various altitudes and fractions of satellite surface area coated. Since equilibrium temperatures decrease with an increase in altitude, the transmitter-unit minimum equilibrium temperature in sunlight was calculated for the maximum expected apogee altitude of 1,600 statute miles and was determined to be 30° C. The values of the emittance of each parallel plate face and the transmitter-unit surface facing the main internal surface of the balloon are shown in table I.

In order to determine what effect an increase or decrease in the percentage of outside surface area coated would have on the transmitter-unit maximum and minimum equilibrium temperatures in sunlight, equation (14) was solved for various altitudes and fractions of outside surface area coated, and the results are presented in figure 9. It may be noted that figure 9 could have been used to determine the design value of 17 percent. The figure indicates that increasing the coating area has less effect in reducing the temperature as the satellite surface area coated becomes greater. In order to determine if the thermal decoupling is sufficient in the earth's shadow, the transmitter-unit maximum and minimum equilibrium temperatures in the earth's shadow were calculated.

Maximum and minimum equilibrium temperatures in earth's shadow.- The derivation of the equations for the transmitter-unit maximum and minimum equilibrium temperatures in the earth's shadow was similar to that in sunlight. The heat balance for the satellite skin adjacent to the transmitter unit is the same as that given by equation (11) with direct earth radiation instead of direct solar radiation. Thus,

$$C_S \frac{1 - a}{4} \alpha_{E,w} K^2 + \sigma \frac{1}{\frac{1}{\epsilon_g} + \frac{1}{\epsilon_g} - 1}\left(T_{B,H}^4 - T_{s,w}^4\right) = \epsilon_{o,w} \sigma T_{s,w}^4 \tag{15}$$

The heat balance for the transmitter unit while the satellite is in the earth's shadow is the same as that for sunlight given by equation (12) with q_3 substituted for q_2. This result yields

$$q_3 + q_1 = \epsilon_d \sigma T_{B,H}^4 + \sigma \frac{1}{\frac{1}{\epsilon_g} + \frac{1}{\epsilon_g} - 1}\left(T_{B,H}^4 - T_{s,w}^4\right) \tag{16}$$

where

$$q_3 = \frac{1}{4} \epsilon_d C_S \frac{1 - a}{2}\left[1 - \left(1 - K^2\right)^{1/2}\right]$$

which is the rate radiation within the main volume of the satellite is absorbed per square centimeter.

Equations (15) and (16) were solved simultaneously to obtain an equation for the transmitter-unit maximum equilibrium temperature in the earth's shadow. This equation is

$$T^4_{B,H} = \frac{\left[\epsilon_g + \epsilon_{o,w}\left(2 - \epsilon_g\right)\right]\left(q_3 + q_1\right) + \epsilon_g C_S \alpha_{E,w} \frac{1 - a}{4} K^2}{\sigma\left[\epsilon_g\left(\epsilon_{o,w} + \epsilon_d - \epsilon_{o,w}\epsilon_d\right) + 2\epsilon_{o,w}\epsilon_d\right]} \qquad (17)$$

Equation (17) can also be used to calculate the transmitter-unit minimum equilibrium temperature in the earth's shadow for various altitudes, simply by putting the term, $\epsilon_g C_S \alpha_{E,w} \frac{1 - a}{4} K^2$, equal to zero in the equation.

The maximum and minimum equilibrium temperatures of the transmitter unit, for sunlight and shadow conditions of the satellite with 17 percent of the outside surface area coated, are presented in figure 10 as a function of satellite altitude. Figure 10 indicates that the shadow temperatures would decay below the minimum allowable temperature of -10° C if the transmitter unit reached thermal equilibrium in the earth's shadow. The thermal response of the transmitter unit is discussed in the following section.

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Thermal response.- Step-by-step calculations were made to determine the stabilized temperature-time curve of the transmitter unit during an orbit passing into the shadow of the earth. Since the Scout vehicle used in placing the satellite into orbit was a developmental vehicle and the 12-foot-diameter inflatable satellite was its first orbital payload, there existed some degree of uncertainty that the nominal orbital parameters would be achieved. For that reason, conservative orbit conditions (that is, unfavorable with regard to heat input) were assumed, namely, an orbit period of 120 minutes with 68-percent sunlight time and 32-percent shadow time, and heating rates corresponding to an altitude of 1,600 statute miles. The 32-percent shadow time in an orbit period of 120 minutes was assumed since the Scout vehicle was not expected to place the satellite in an orbit having a period greater than 120 minutes, and a separate analysis indicated that the maximum time in shadow would be 32 percent of the orbit period. Since heating rates decrease with increasing altitude, the heating rates corresponding to a maximum expected apogee altitude of 1,600 statute miles were assumed for the analysis of the thermal response.

The equation for the thermal response of the transmitter unit is

$mc_{p,B} \frac{\Delta T_B}{\Delta t}$ = Reception from main internal surface of balloon

+ Reception across radiative plate gap + Internal heat dissipation

- Radiation across radiative plate gap - Radiation from transmitter-unit surface facing main internal volume of balloon

By utilizing this equation the step-by-step calculations were made around several orbits to determine the stabilized temperature-time curve of the transmitter unit during an orbit passing into the shadow of the earth. The temperatures of the transmitter unit throughout the stabilized cycle are presented in figure 11. A comparison of figures 6 and 11 shows the advantage of the thermal decoupling gap, for while the satellite skin temperature is oscillating between -117° C and 11° C, the transmitter-unit temperature oscillates between 9° C and 22° C. It should be noted that, to simplify the calculations, the input from the earth-reflected sunlight was assumed constant during the sunlit part of the orbit, and equal to its maximum value. The inaccuracy due to this simplification is not very large, since if this reflected sunlight is neglected altogether the calculated temperature of the transmitter unit oscillates between 4° C and 15° C instead of between 9° C and 22° C.

Thermal Conduction

The thermal decoupling shield was designed such that conduction along the attachments from the satellite skin to the transmitter and battery units would be negligible. However, the nitrogen gas used in inflating the satellite could thermally couple the transmitter and battery units with the satellite skin during the period before the gas leaked out. For this reason, a 100-percent sunlight orbit would have been desirable for the first several days in order to reduce the possibility of freezing the units. For such an orbit, the temperature of the beacon units would be very close to the average temperature of the satellite skin, and conduction through the nitrogen gas would have little effect on the thermal balance. However, information indicated that the batteries would not be permanently damaged if they froze. Therefore, the effect of the inflation gas was neglected since calculations indicated that, because of the leak provided, the internal gas pressure in the satellite would probably be below a value of 10^{-4} millimeters of mercury in 18 hours. The value of 10^{-4} millimeters of mercury was determined as the value at which radiation becomes the only significant means of heat transfer.

Because the inflation gas in the satellite would thermally couple the transmitter and battery units with the average temperature of the satellite skin in sunlight and in the earth's shadow, the radiation-balance equations previously discussed would not be valid until the inflation gas leaked out of the satellite.

RESULTS AND DISCUSSION

The Explorer IX, an inflatable spherical satellite having a 12-foot diameter, was injected into orbit at 13:16 Greenwich mean time (GMT) on February 16, 1961 from NASA Wallops Station. Figure 12 shows the earth track during the launch, injection, and first orbital pass. The beacon was first received by the Minitrack station at Esselen Park, South Africa at 13:44 GMT, and subsequently tracked until 14:06 GMT. This tracking interval is shown in figure 12. The beacon was received next by the Minitrack station at Woomera, Australia from 14:20 GMT to 14:41 GMT. L
The tracking interval for Woomera is also shown in figure 12. As noted 1
in the figure, the satellite entered into the earth's shadow at 14:24 GMT 9
and was, therefore, tracked by Woomera for 17 minutes while in shadow. 5
This was the last time the tracking beacon was received. The next oper- 2
ating Minitrack station was at San Diego, California, but San Diego failed to acquire the Explorer IX beacon. The satellite would have been in sunlight 4 or 5 minutes when the San Diego Minitrack station should have acquired the beacon. The Blossom Point, Maryland, Minitrack station also failed to acquire the satellite beacon. The Minitrack stations were then instructed to switch their tracking to the frequency of the beacon located on the fourth stage of the Scout launching vehicle, and to stand by on the frequency of the satellite beacon.

Reception of the satellite beacon by the Minitrack stations indicated that the radio tracking beacon transmitter set had survived the ejection and inflation of the 12-foot-diameter satellite. Reception also indicated that the unique use of the satellite skin as the transmitting dipole antenna by means of the insulating equatorial band was successful.

When the satellite finally entered the earth's shadow it had been in sunlight for 68 minutes and was still operating properly. The first 68 minutes of orbit while the satellite was in sunlight was sufficient time for the transmitter and battery units to reach thermal equilibrium at approximately the average skin temperature since the satellite skin would reach thermal equilibrium in approximately 10 minutes as was shown in figure 6. Without the coating of white epoxy enamel, the temperature would have been about 90° C (see fig. 4 for altitudes of 400 to 600 statute miles during injection and subsequent 20-minute period) which is well above acceptable operating temperatures. The proper operation of the transmitter set for this period of time indicated that the satellite temperature did not exceed the maximum value for satisfactory operation of the transmitter set. Therefore, the white epoxy enamel coating over 17 percent of the balloon outside surface gave a satisfactory temperature control for the radio beacon set in sunlight.

Since the inflating gas was present when the satellite entered the earth's shadow, the conduction through the nitrogen gas could cause the temperatures of the transmitter and battery units to decrease to about -40° C to -50° C or lower near the end of the time in shadow. The heat transfer due to gas conduction would be further induced because the satellite is spinning at approximately 30 revolutions per minute. The time of 14:41 GMT at which the Minitrack station at Woomera ceased to receive the Explorer IX beacon was determined as that at which the Explorer IX satellite passed below the horizon. Two explanations for beacon reception under the low-temperature conditions are possible. First, although the transistors and storage-battery cells of the radio beacon set operate most favorably in a temperature range from -10° C to 60° C, and operation is critical above 60° C, they could operate at temperatures as low as -30° C for a short period of time. Second, the internal-heat dissipation inside the transmitter unit could hinder the rate at which the unit approached the average equilibrium temperature of the satellite skin in the earth's shadow. The hindrance would be possible since the internal heat dissipated must be conducted to the surfaces of the transmitter unit to be emitted and conducted away by the nitrogen gas. In any case, the low average temperature in the earth's shadow would not destroy the transmitter and battery units as was already mentioned.

The failure of the radio beacon set has been attributed to over-voltage on the transistors when the satellite reentered sunlight approximately 5 minutes before coming within range of the San Diego Minitrack station. Since the radio beacon was acquired while the satellite was in the shadow of the earth and the transmitter is powered by storage batteries during this time, it may be concluded that the battery switch performed satisfactorily thereby connecting the batteries to the transmitter circuit. Although the batteries act as over-voltage protectors since the excess electrical energy from the solar-cells power supply is converted into heat in the batteries, this function would no longer be performed by the batteries if they became extremely cold. During the time the inflation gas is present the batteries would become extremely cold in the shadow of the earth as previously mentioned, and their internal resistance would increase rapidly with decreasing temperature near -40° C as has been shown in a separate study (fig. 13). Solar cells are known to have a higher output voltage at the low temperatures which would be acquired in the earth's shadow. While the satellite was in the earth's shadow, the decreased temperature of the battery supply resulted in a very high battery resistance, effectively removing the battery load from the solar cells. The solar-cell terminal voltage thus increased because of the decreased load in addition to the higher voltage brought about by their higher efficiency at lower temperatures. As a result an excessive voltage would be applied to the transmitter. This high voltage probably damaged the transistors in the transmitter, effectively destroying the transmitter. Figure 1 shows that there is no over-voltage protector in the transmitter circuit other than the batteries; this type of protection was overlooked in the electronic design of the transmitter. A simple Zener diode voltage regulator would have eliminated this type of transmitter failure without excessive rework.

CONCLUSIONS

An analytical study was made to determine a method of maintaining the temperature of a radio tracking beacon on a 12-foot-diameter inflatable satellite (Explorer IX) within acceptable limits. From this study and from the behavior of the beacon after the satellite was put in orbit, it is concluded that:

1. Calculations indicate that proper temperature control can be
obtained in orbit for the radio-beacon-set components by applying a L
coating having a low ratio of solar absorptance to emittance to a per- 1
centage of the balloon outside surface area for control while the satel- 9
lite is in sunlight and by physically separating the beacon transmitter 5
and battery units from the balloon inside surface in order to minimize 2
the rate of heat transfer for control while the satellite is in the earth's shadow. The analysis indicates that the emittance of the inside surface of the balloon should be high (plastic film rather than aluminum) to minimize the temperature difference over the balloon surface. Therefore, radio tracking beacon transmitter sets can be kept within finite temperature limits for application to inflatable satellites which have a low mass per unit area, such as Explorer IX.

2. The proper operation of the transmitter for the 68-minute period of time in sunlight indicates that by coating the satellite with white epoxy enamel, the satellite temperature did not exceed the maximum value for satisfactory operation of the radio beacon set.

3. The most probable cause of transmitter failure is excessive voltage applied to the transmitter, with consequent destruction of the transistors. The failure occurred during the first orbit before the inflation gas leaked out of the satellite. During this time the inflation gas provided increased heat transfer between the satellite skin and the transmitter and battery units. The temperatures of the satellite skin and the radio beacon set became very low when the satellite was in the earth's shadow; then on entry of the satellite into sunlight, higher than normal voltage from the cold solar cells and high internal resistance of the cold batteries caused excessive voltage to be applied to the transmitter.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., April 26, 1962.

APPENDIX

THERMAL RESPONSE OF THE 12-FOOT-DIAMETER INFLATABLE SATELLITE

The differential equation for the thermal response of the 12-foot-diameter inflatable satellite is

$$Mc_{p,s} \frac{dT}{dt} = \text{Rate of heat gain} - \text{Rate of heat loss} \tag{A1}$$

The satellite has a laminated skin construction of four alternate layers of plastic film and aluminum foil. The thicknesses of the laminated skin layers are as follows:

Outside layer = 0.50-mil-thick aluminum foil

Second layer = 0.50-mil-thick plastic film

Third layer = 0.50-mil-thick aluminum foil

Inside layer = 0.50-mil-thick plastic film

The specific heat was assumed as a mean specific heat for the laminated skin construction.

The density and specific heat of the plastic film were obtained from the manufacturer as

$$\rho_{my} = 1.38 \text{ grams/cm}^3$$

$$c_{p,my} = -0.0114 + 0.000935T \qquad (203^\circ \text{ K} \leqq T \leqq 358^\circ \text{ K})$$

The density of aluminum is 2.70 grams/cm^3. The specific heat of aluminum, obtained from reference 9, was approximated as two straight-line functions, as follows:

$$c_{p,A} = 0.0696 + 0.000560909T \qquad (130^\circ \text{ K} \leqq T \leqq 220^\circ \text{ K})$$

$$c_{p,A} = 0.1390 + 0.000244722T \qquad (220^\circ \text{ K} \leqq T \leqq 360^\circ \text{ K})$$

The error in this approximation is less than $1\frac{1}{4}$ percent.

Thus, the specific heat of the satellite is

$$c_{p,s} = 0.04220 + 0.00068744T \qquad (203^\circ\ K \leqq T \leqq 220^\circ\ K)$$

$$c_{p,s} = 0.08813 + 0.00047820T \qquad (220^\circ\ K \leqq T \leqq 358^\circ\ K)$$

Shadow to Sunlight

The differential equation for the temperature of the skin of the 12-foot-diameter satellite in sunlight is (see eq. (A1))

$$Mc_{p,s}\frac{dT}{dt} = \text{Rate of heat gain} - \text{Rate of heat loss}$$

Substituting for the first term on the right, the maximum energy reception in sunlight (ref. 5), yields

$$Mc_{p,s}\frac{dT}{dt} = \frac{\pi D^2}{4} C_S\alpha_S\left\{1 + 2\left(a + \frac{1 - a}{4}\frac{\alpha_E}{\alpha_S}\right)\left[1 - \left(1 - K^2\right)^{1/2}\right]\right\} - \pi D^2\epsilon_o\sigma\overline{T^4} \tag{A2}$$

Dividing through by $Mc_{p,s}$ yields

$$\frac{dT}{dt} = \frac{\frac{\pi D^2}{4M} C_S\alpha_S\left\{1 + 2\left(a + \frac{1 - a}{4}\frac{\alpha_E}{\alpha_S}\right)\left[1 - \left(1 - K^2\right)^{1/2}\right]\right\} - \frac{\pi D^2}{M}\epsilon_o\sigma\overline{T^4}}{c_{p,s}} \tag{A3}$$

Let

$$\frac{dT}{dt} = \frac{X^2 - Y^2\overline{T^4}}{V^2 + W^2T} \tag{A4}$$

where:

$$X^2 = \frac{\pi D^2}{4M} C_S\alpha_S\left\{1 + 2\left(a + \frac{1 - a}{4}\frac{\alpha_E}{\alpha_S}\right)\left[1 - \left(1 - K^2\right)^{1/2}\right]\right\}$$

$$y^2 = \frac{\pi D^2}{M} \epsilon_0 \sigma$$

$$V^2 = \begin{cases} 0.04220 & (203^\circ \text{ K} \leqq T \leqq 220^\circ \text{ K}) \\ 0.08813 & (220^\circ \text{ K} \leqq T \leqq 358^\circ \text{ K}) \end{cases}$$

$$W^2 = \begin{cases} 0.00068744 & (203^\circ \text{ K} \leqq T \leqq 220^\circ \text{ K}) \\ 0.00047820 & (220^\circ \text{ K} \leqq T \leqq 358^\circ \text{ K}) \end{cases}$$

By the use of partial fractions, equation (A4) was integrated to yield an equation for the thermal response in sunlight:

$$t - t_0 = \frac{V^2}{4x\sqrt{xy}}\left(\log \frac{\sqrt{x/y} + \overline{T}}{\sqrt{x/y} - \overline{T}} - \log \frac{\sqrt{x/y} + \overline{T_0}}{\sqrt{x/y} - \overline{T_0}} + 2 \tan^{-1} \frac{\overline{T}}{\sqrt{x/y}} - 2 \tan^{-1} \frac{\overline{T_0}}{\sqrt{x/y}}\right)$$

$$+ \frac{W^2}{4xy}\left(\log \frac{x/y + \overline{T}^2}{x/y - \overline{T}^2} - \log \frac{x/y + \overline{T_0}^2}{x/y - \overline{T_0}^2}\right) \qquad \text{(A5)}$$

where

$\overline{T_0}$ average equilibrium skin temperature at a time $t_0 = 0$ when the satellite emerges from the earth's shadow

$\overline{T}$ average skin temperature at any time t in sunlight

For simplicity, no differentiation was made between $\overline{T}$ and $\left(\overline{T^4}\right)^{1/4}$ since each would be approximately the same.

Sunlight to Shadow

The differential equation for the temperature of the skin of the 12-foot-diameter satellite in the earth's shadow is (see eq. (A1))

$$Mc_{p,s} \frac{dT}{dt} = \text{Rate of heat gain} - \text{Rate of heat loss}$$

Substituting for the first term on the right, the energy reception in the shadow of the earth (ref. 5), yields

$$Mc_{p,s}\frac{dT}{dt} = \frac{\pi D^2}{4} C_S \alpha_E \frac{1 - a}{2}\left[1 - \left(1 - K^2\right)^{1/2}\right] - \pi D^2 \epsilon_o \sigma \overline{T^4} \tag{A6}$$

Dividing through by $Mc_{p,s}$ yields

$$\frac{dT}{dt} = \frac{\frac{\pi D^2}{4M} C_S \alpha_E \frac{1 - a}{2}\left[1 - \left(1 - K^2\right)^{1/2}\right] - \frac{\pi D^2}{M} \epsilon_o \sigma \overline{T^4}}{c_{p,s}} \tag{A7}$$

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Let

$$\frac{dT}{dt} = \frac{z^2 - y^2\overline{T^4}}{v^2 + w^2 T} \tag{A8}$$

where:

$$z^2 = \frac{\pi D^2}{4M} C_S \alpha_E \frac{1 - a}{2}\left[1 - \left(1 - K^2\right)^{1/2}\right]$$

By the use of partial fractions, equation (A8) was integrated to yield an equation for the thermal response in the shadow of the earth:

$$t - t_0 = \frac{v^2}{4z\sqrt{yz}}\left(-\log\frac{\overline{T} - \sqrt{z/y}}{\overline{T} + \sqrt{z/y}} + \log\frac{\overline{T}_0 - \sqrt{z/y}}{\overline{T}_0 + \sqrt{z/y}} + 2\tan^{-1}\frac{\overline{T}}{\sqrt{z/y}} - 2\tan^{-1}\frac{\overline{T}_0}{\sqrt{z/y}}\right)$$

$$+ \frac{w^2}{4yz}\left(-\log\frac{\overline{T}^2 - z/y}{\overline{T}^2 + z/y} + \log\frac{\overline{T}_0^2 - z/y}{\overline{T}_0^2 + z/y}\right) \tag{A9}$$

where

$\overline{T_0}$ average equilibrium skin temperature at a time $t_0 = 0$ when the satellite enters the earth's shadow (assumed as that resulting from maximum solar reflected radiation)

$\overline{T}$ average skin temperature at any time t in the earth's shadow

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REFERENCES

1. Whitney, Charles A.: The Structure of the High Atmosphere - I. Linear Models. Ephemeris of Satellite 1957 Alpha 2 and Collected Reports on Satellite Observations. IGY Satellite Rep. Ser. No. 8, Nat. Acad. Sci., June 15, 1959, pp. 103-114.

2. Whitney, Charles A.: The Structure of the High Atmosphere - II. A Conduction Model. Ephemeris of Satellite 1957 Alpha 2 and Collected Reports on Satellite Observations. IGY Satellite Rep. Ser. No. 8, Nat. Acad. Sci., June 15, 1959, pp. 115-121.

3. King-Hele, D. G., Cook, G. E., and Walker, D. M. C.: The Contraction of Satellite Orbits Under the Influence of Air Drag. Part I. With Spherically Symmetrical Atmosphere. Tech. Note No. G. W. 533, British R.A.E., Nov. 1959.

4. Coffee, Claude W., Jr., Bressette, Walter E., and Keating, Gerald M.: Design of the NASA Lightweight Inflatable Satellites for the Determination of Atmospheric Density at Extreme Altitudes. NASA TN D-1243, 1962.

5. Wood, George P., and Carter, Arlen F.: Predicted Characteristics of an Inflatable Aluminized-Plastic Spherical Earth Satellite With Regard to Temperature, Visibility, Reflection of Radar Waves, and Protection From Ultraviolet Radiation. NASA TN D-115, 1959.

6. Cunningham, F. G.: Power Input to a Small Flat Plate From a Diffusely Radiating Sphere, With Application to Earth Satellites. NASA TN D-710, 1961.

7. Anon.: Radio Beacon Set for 12-Foot Inflatable Satellite. AED 404 (Contract No. NAS 1-863), Radio Corp. of America, Mar. 1961.

8. Kreith, Frank: Principles of Heat Transfer. Int. Textbook Co. (Scranton, Pa.), c.1958, pp. 175-232.

9. Arne, V. L., and Wolfe, G. W.: Thermal Properties of Solids. Rep. No. E9R-12073, Vought Astronautics, Chance Vought Aircraft, Inc., Oct. 13, 1959.

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TABLE I

RADIATION PROPERTIES OF SURFACES

Surface	Emittance, ϵ	Absorptance, α	α/ϵ
Aluminum-foil surface of laminated material	0.08	0.15	1.88
Plastic-film surface of laminated material	0.5	----	----
White-epoxy-enamel surface	0.94	0.33	0.35
Transmitter surface facing main internal surface of satellite	0.15	----	----
Thermal-gap surfaces	0.08	----	----

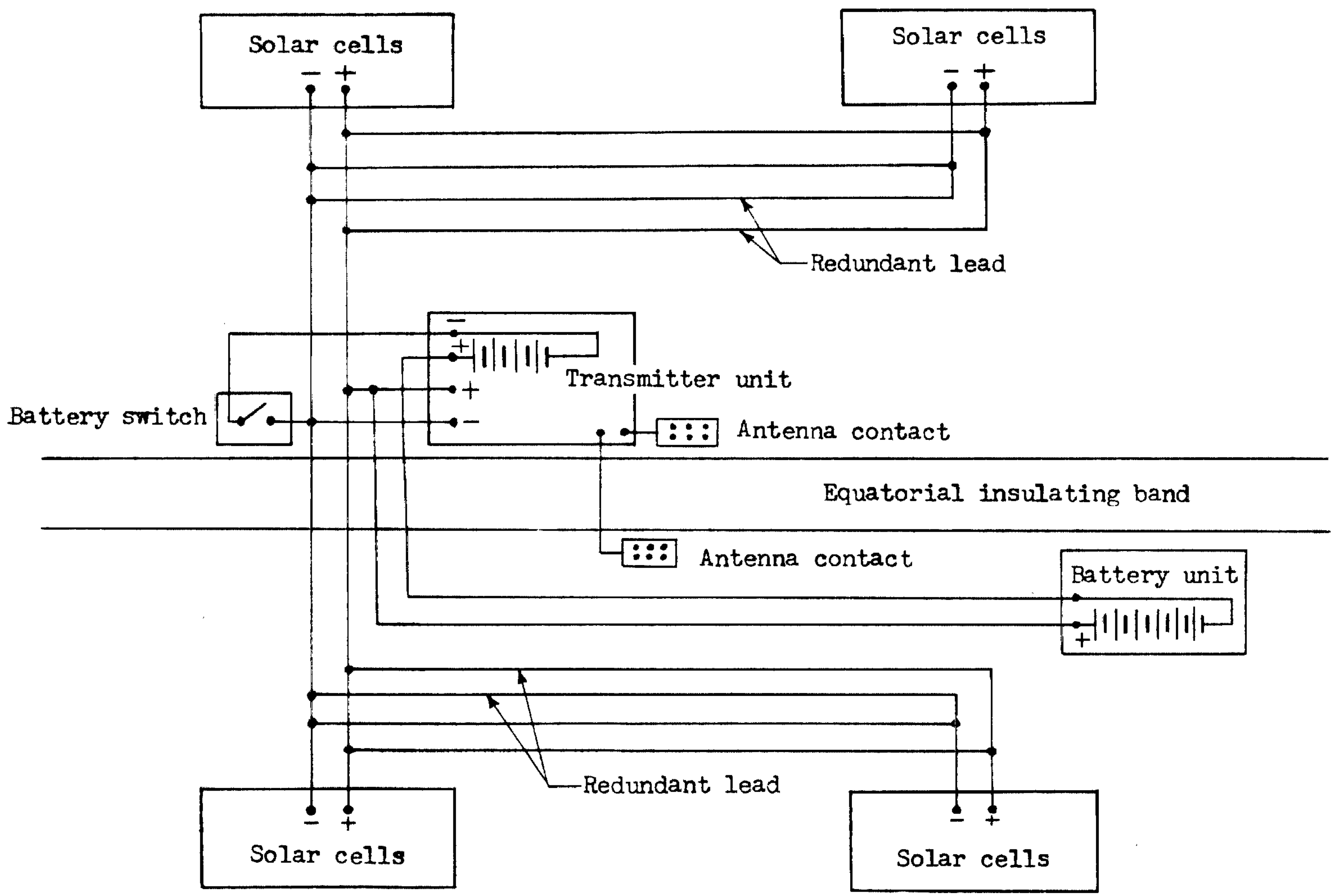
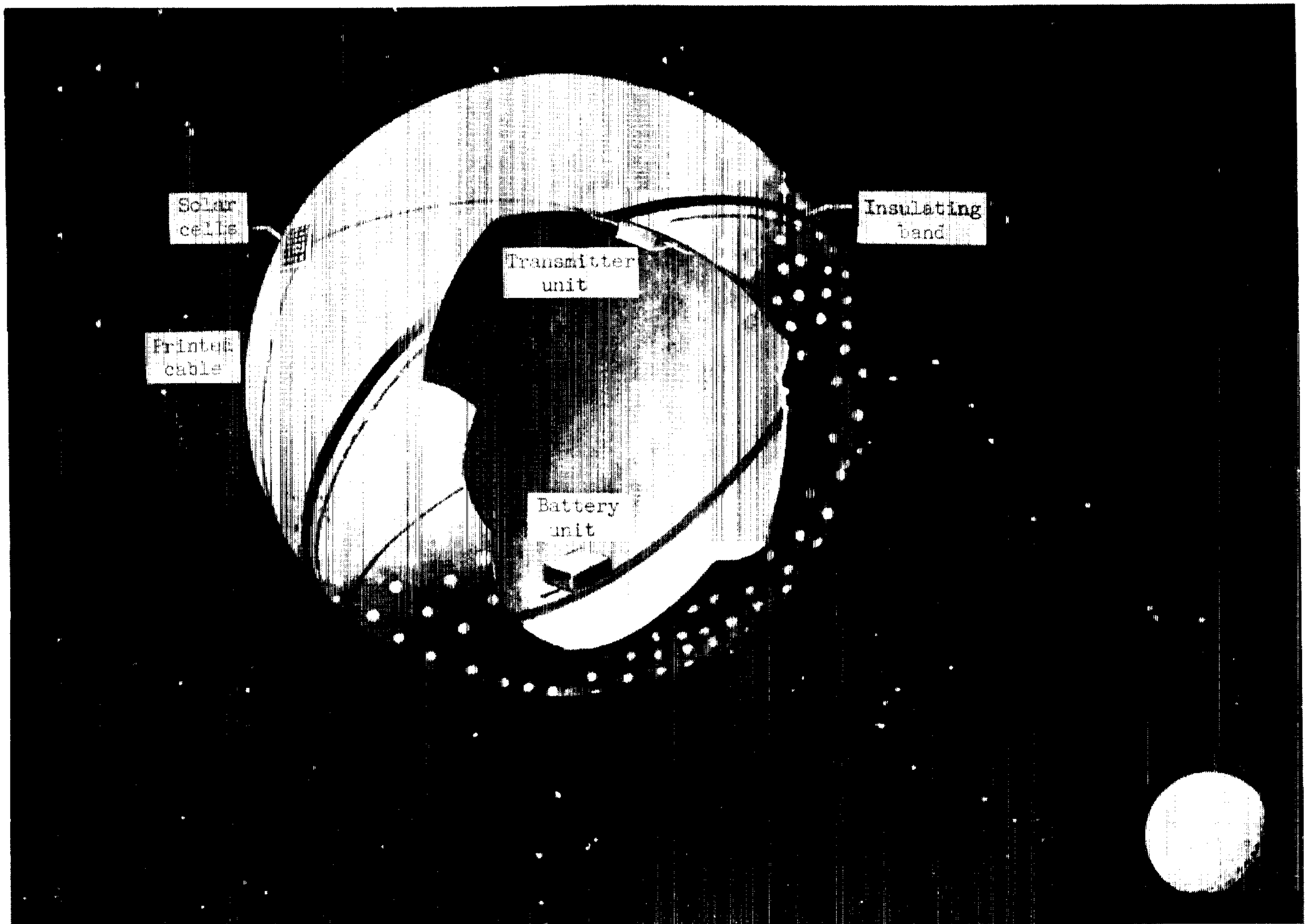


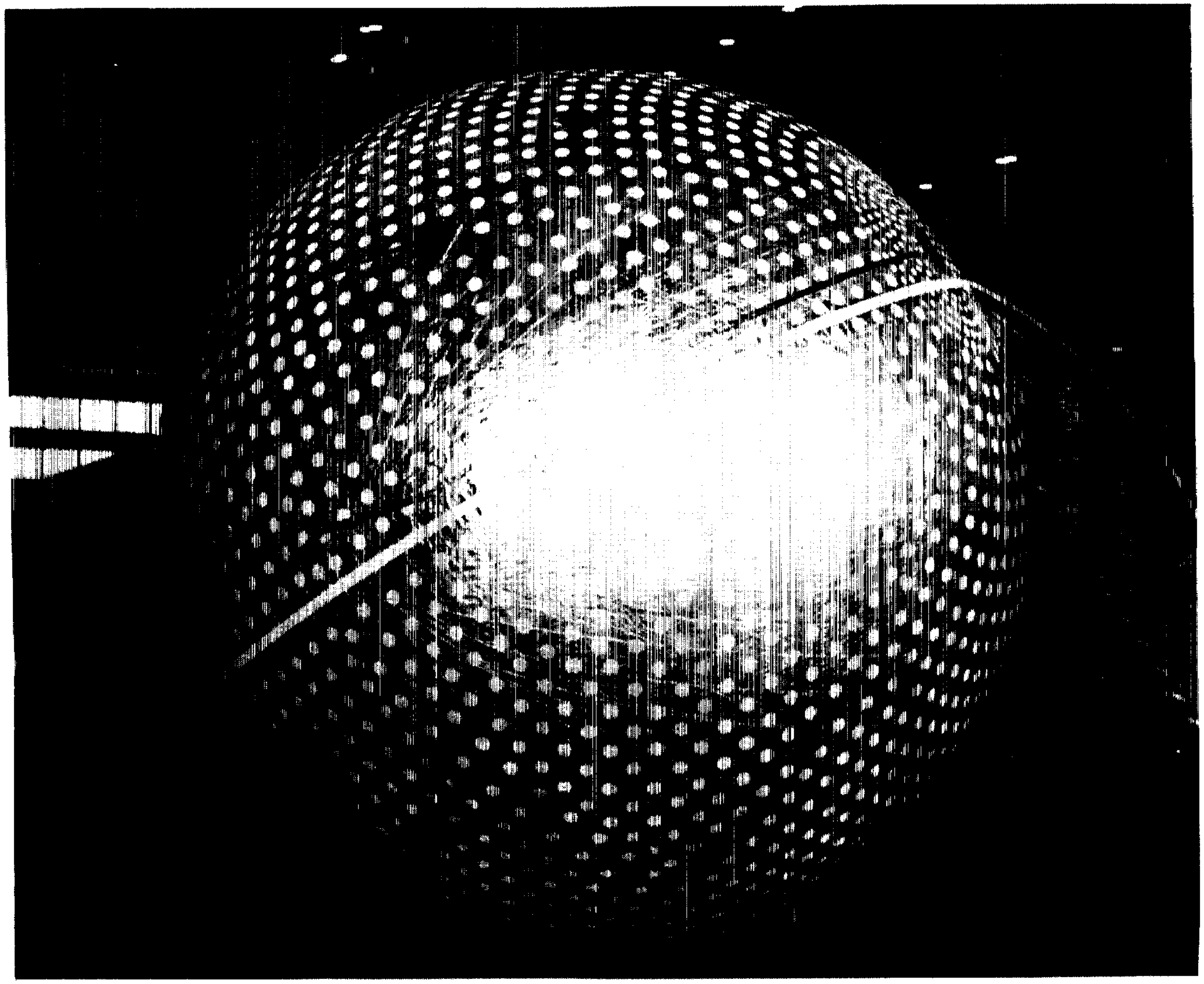
Figure 1.- Wiring diagram for radio tracking beacon transmitter system.

L-1952



L-60-7809.1

Figure 2.- Cutaway drawing of the 12-foot-diameter satellite in orbit.



L-61-3762

Figure 3.- Photograph of the 12-foot-diameter satellite after application of white epoxy enamel.

L-1952

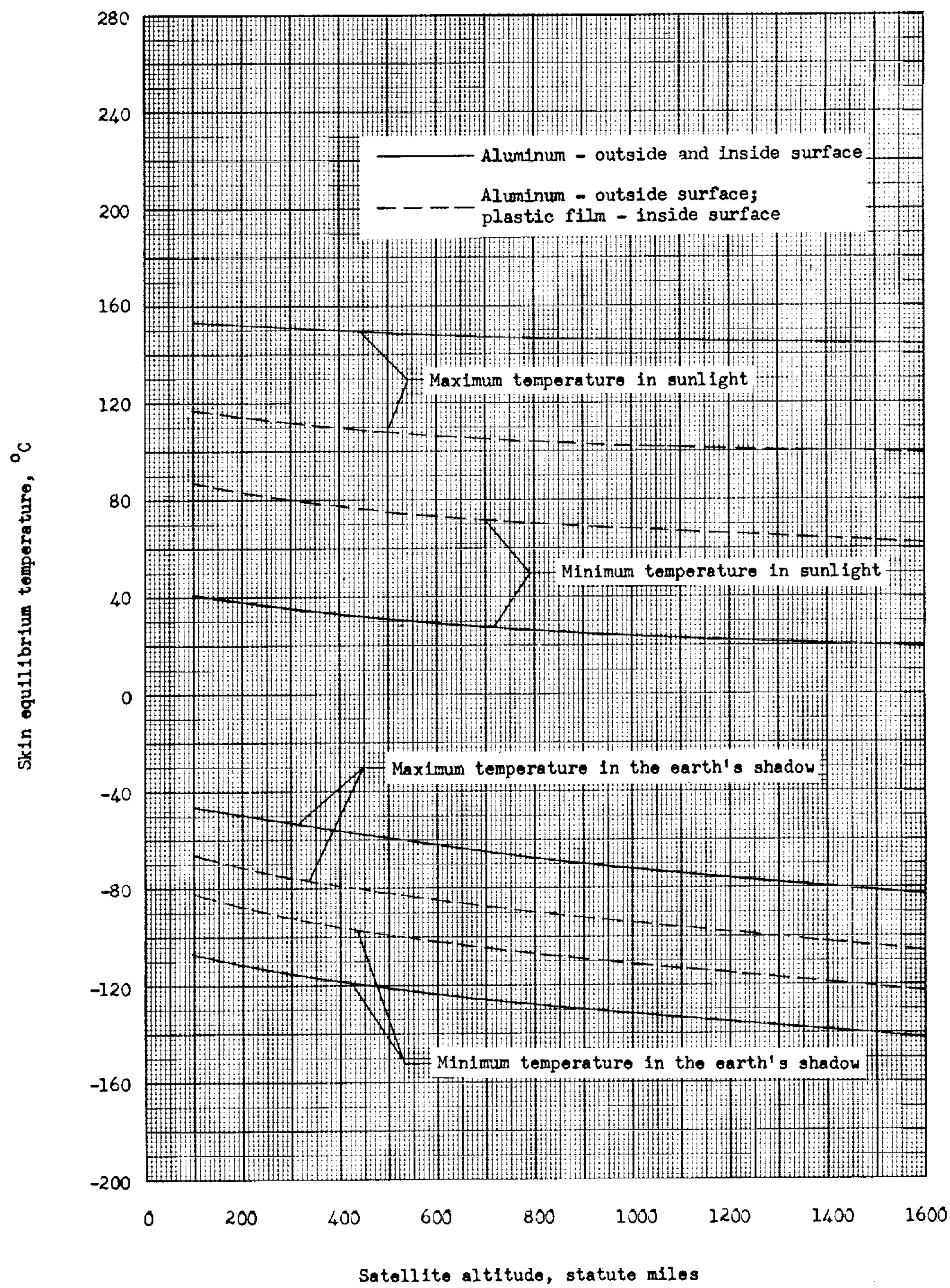


Figure 4.- Variation of skin equilibrium temperatures with satellite altitude for two laminate skin constructions for the 12-foot-diameter satellite.

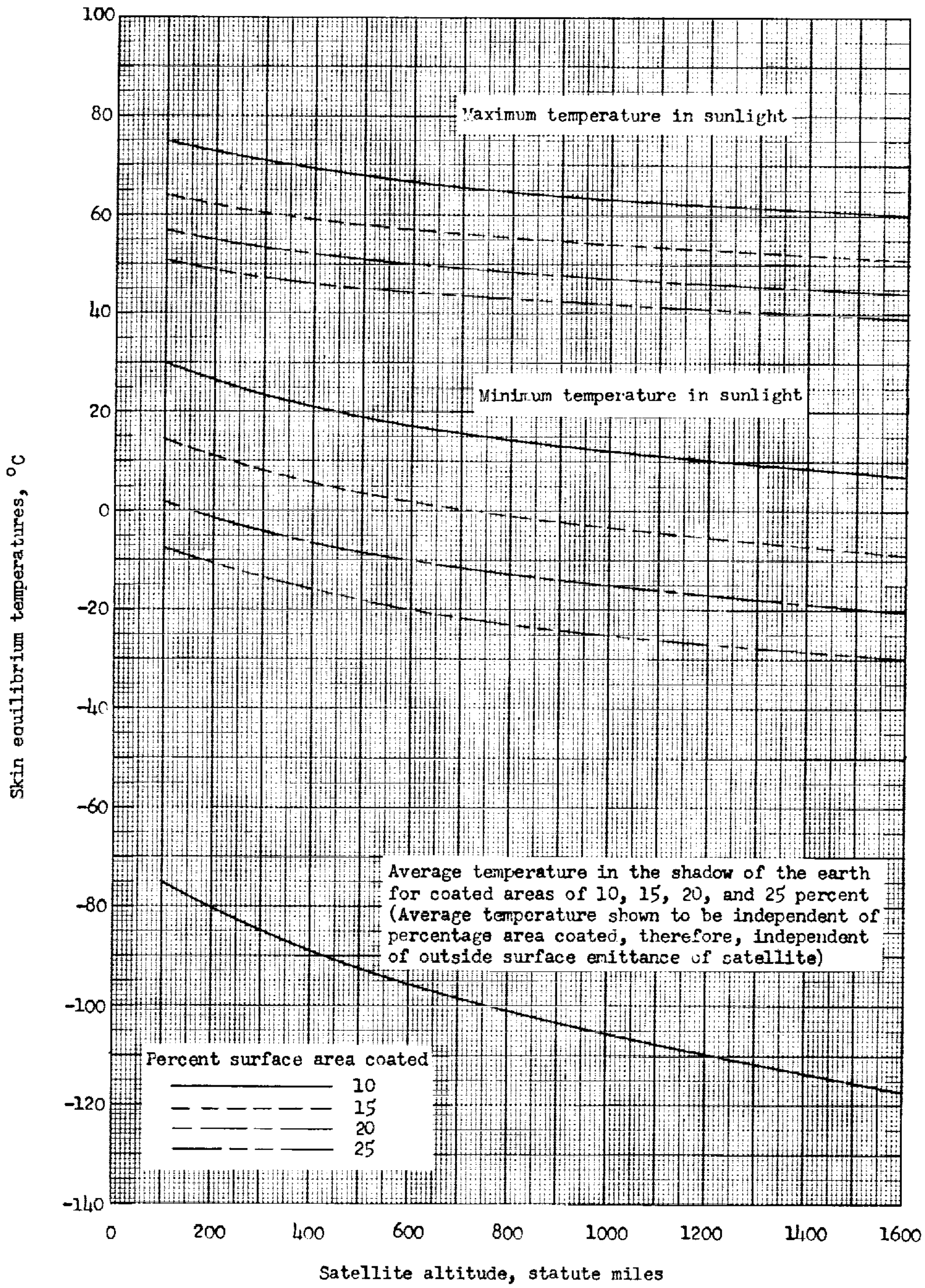


Figure 5.- Variation of skin equilibrium temperatures with satellite altitude for the 12-foot-diameter satellite for 10, 15, 20, and 25 percent of the outside surface area coated with white epoxy enamel.

L-1952

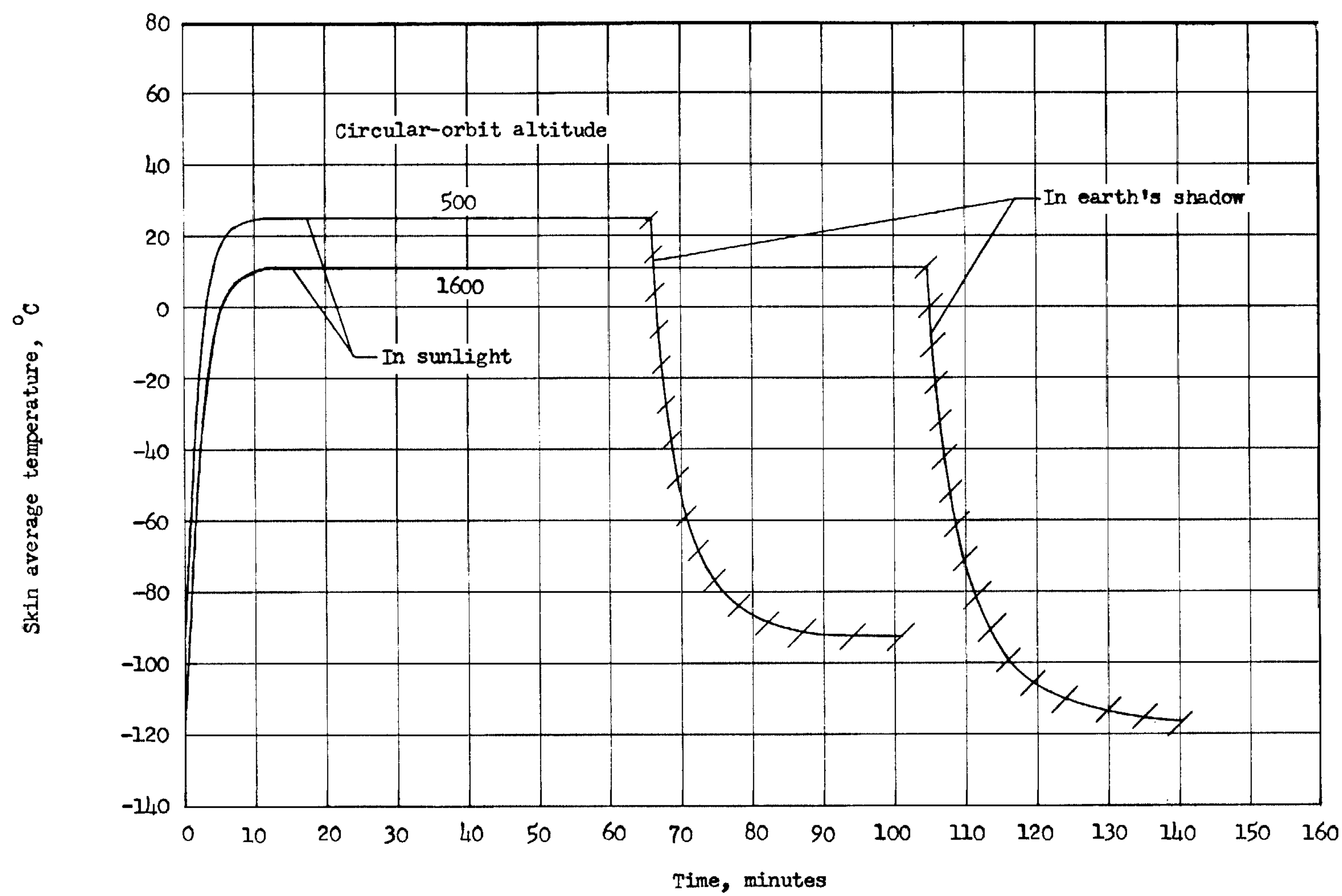


Figure 6.- Thermal response of the skin of the 12-foot-diameter satellite with 17 percent of the outside surface area coated with white epoxy enamel at circular-orbit altitudes of 500 and 1,600 statute miles.

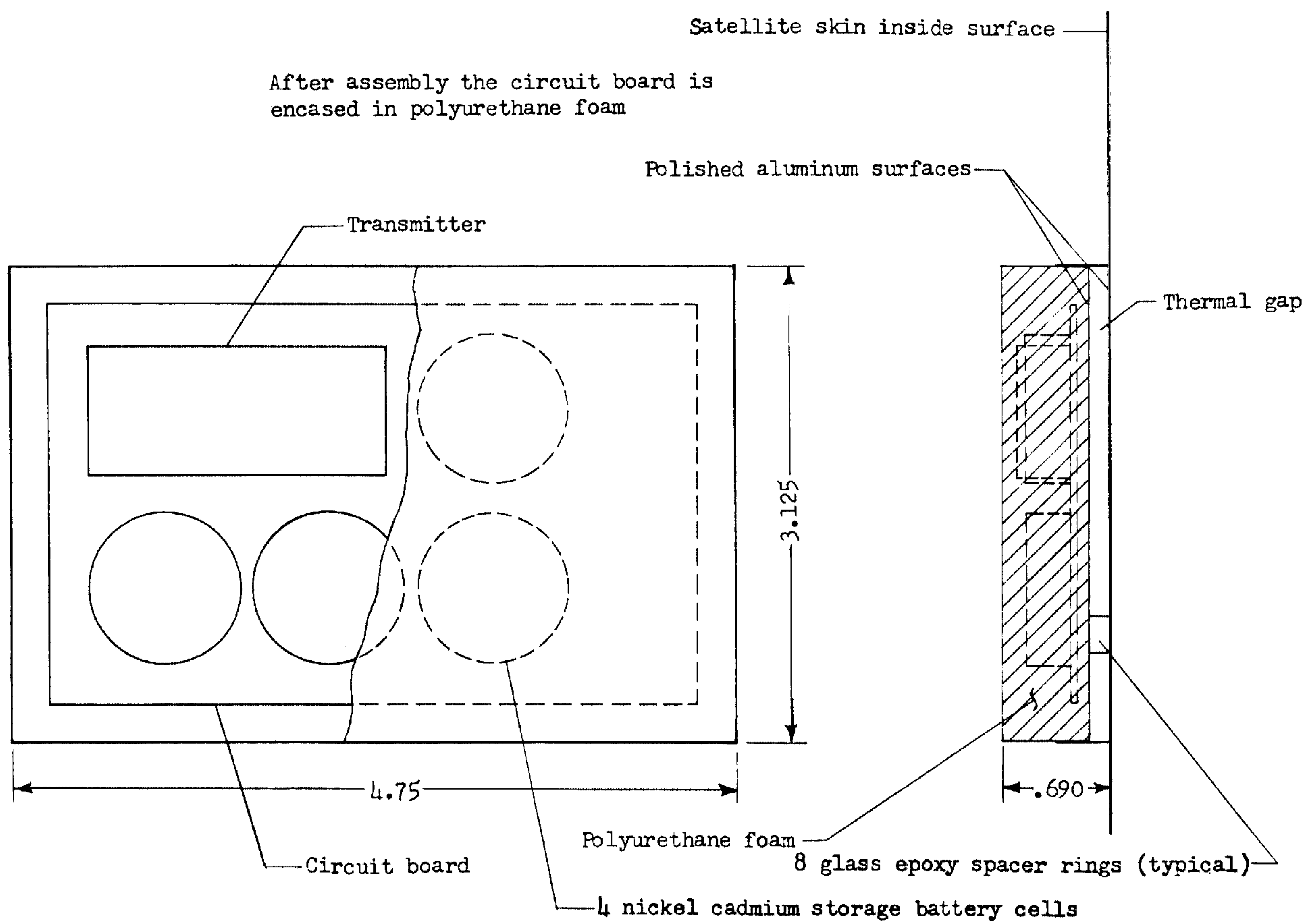
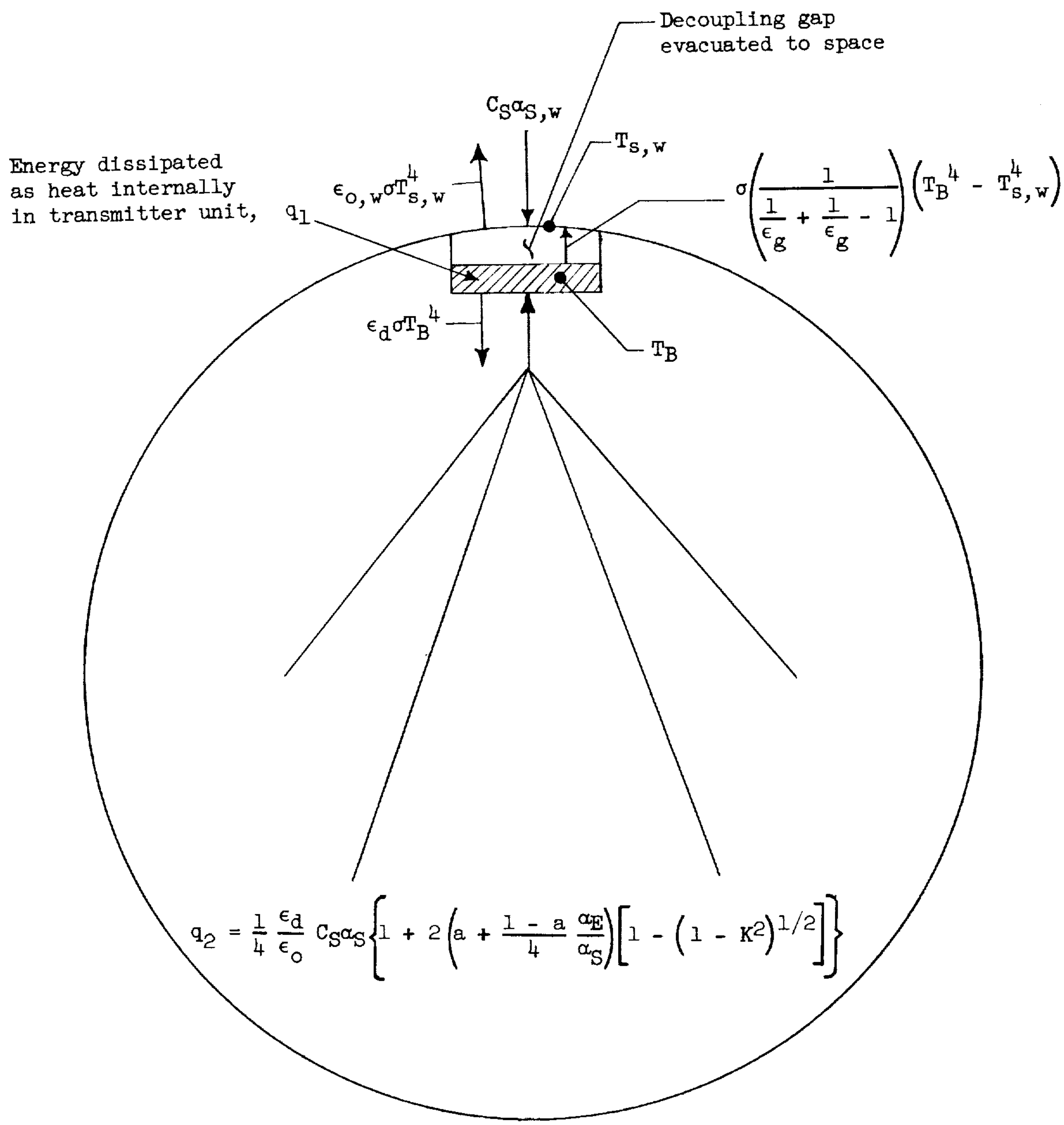


Figure 7.- General arrangement of transmitter unit.



L-1952

Figure 8.- Schematic diagram with different components of the thermal balance in sunlight for the transmitter unit and the skin of the 12-foot-diameter inflatable satellite.

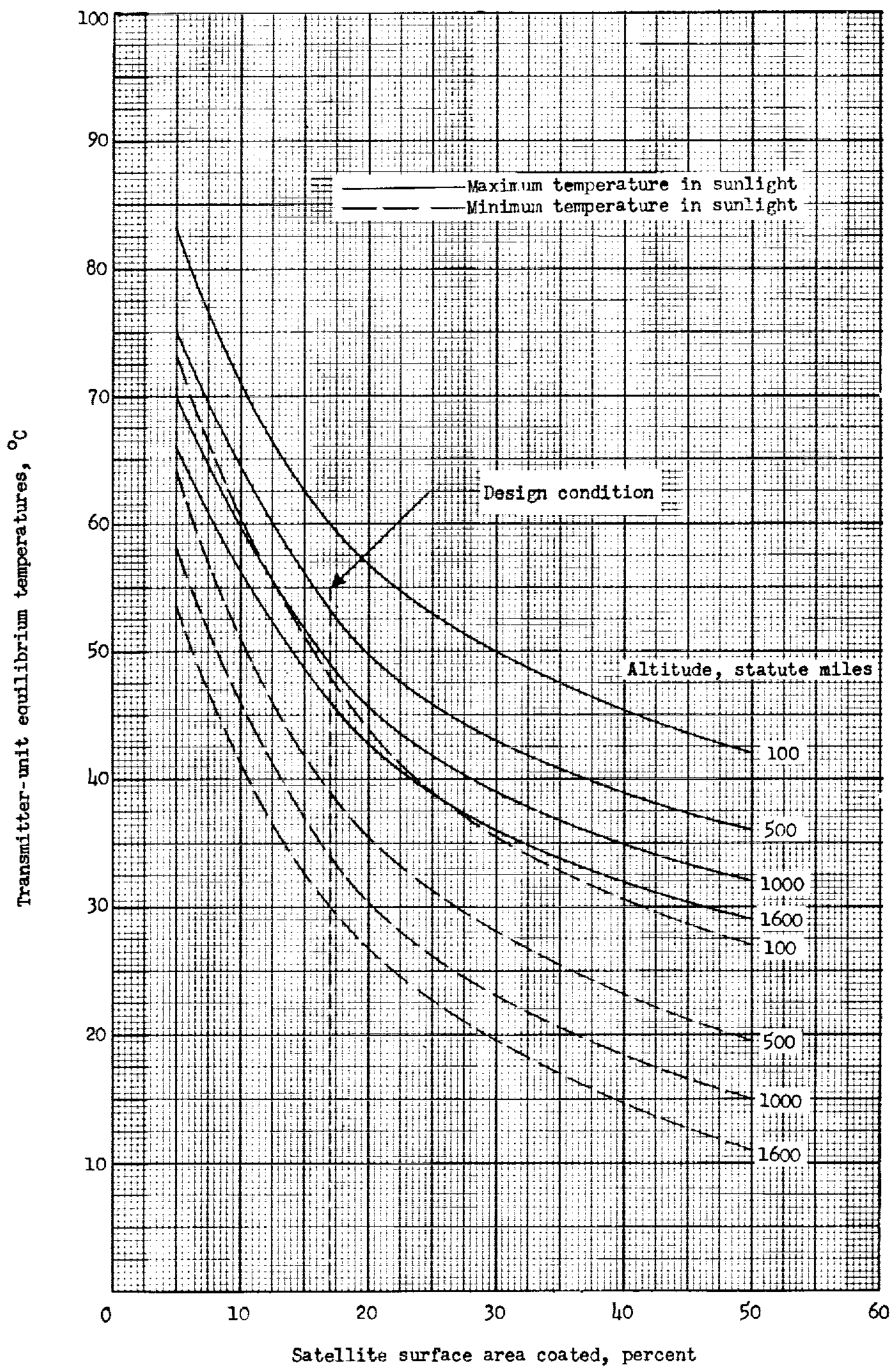


Figure 9.- Variation of transmitter-unit equilibrium temperatures in sunlight with percentage of satellite outside surface area coated with white epoxy enamel for several circular orbit altitudes.

L-1592

L-1952

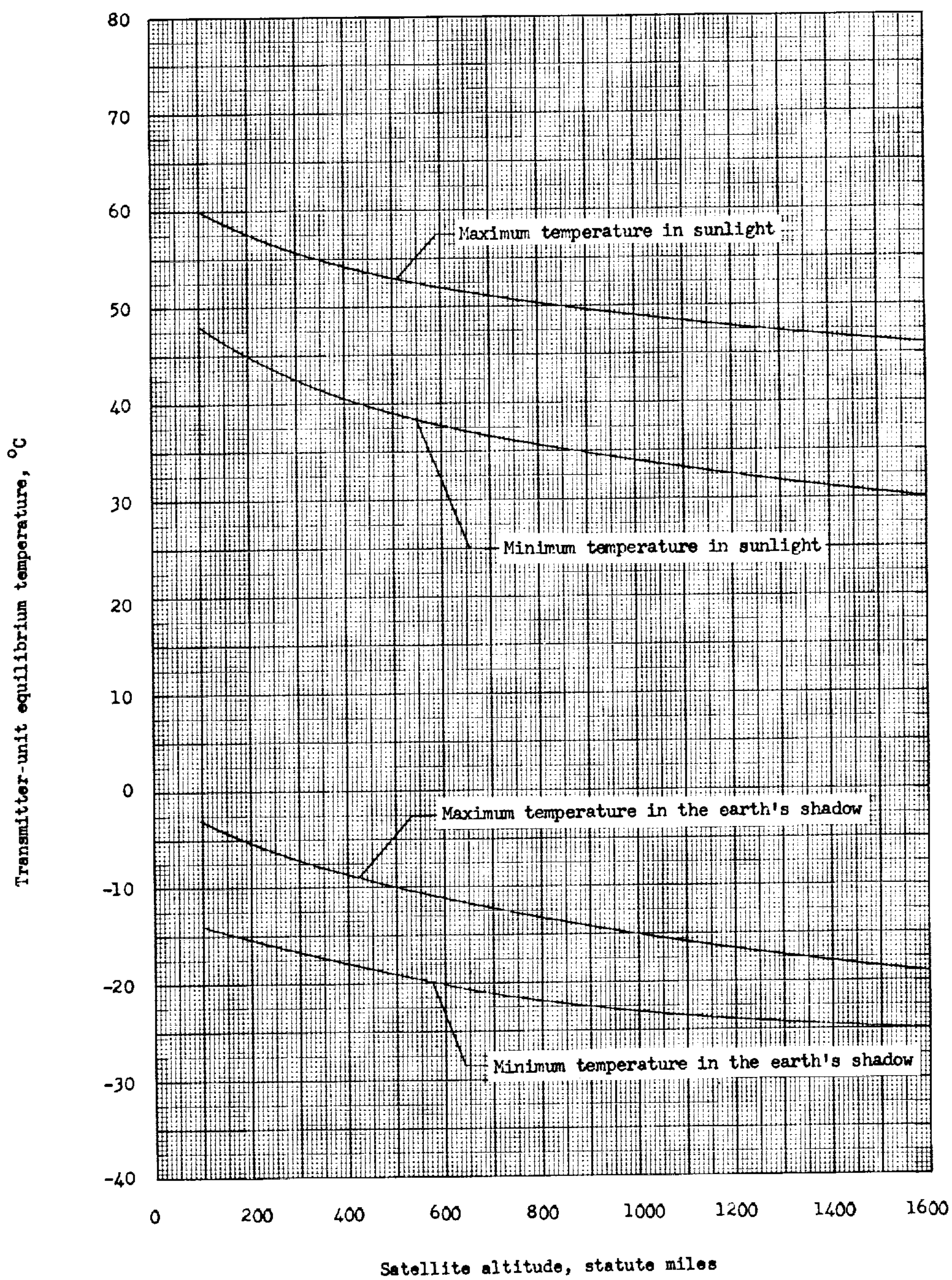


Figure 10.- Variation with altitude of transmitter-unit maximum and minimum equilibrium temperatures in sunlight or the earth's shadow for the 12-foot-diameter satellite with 17 percent of the outside surface area coated with white epoxy enamel.

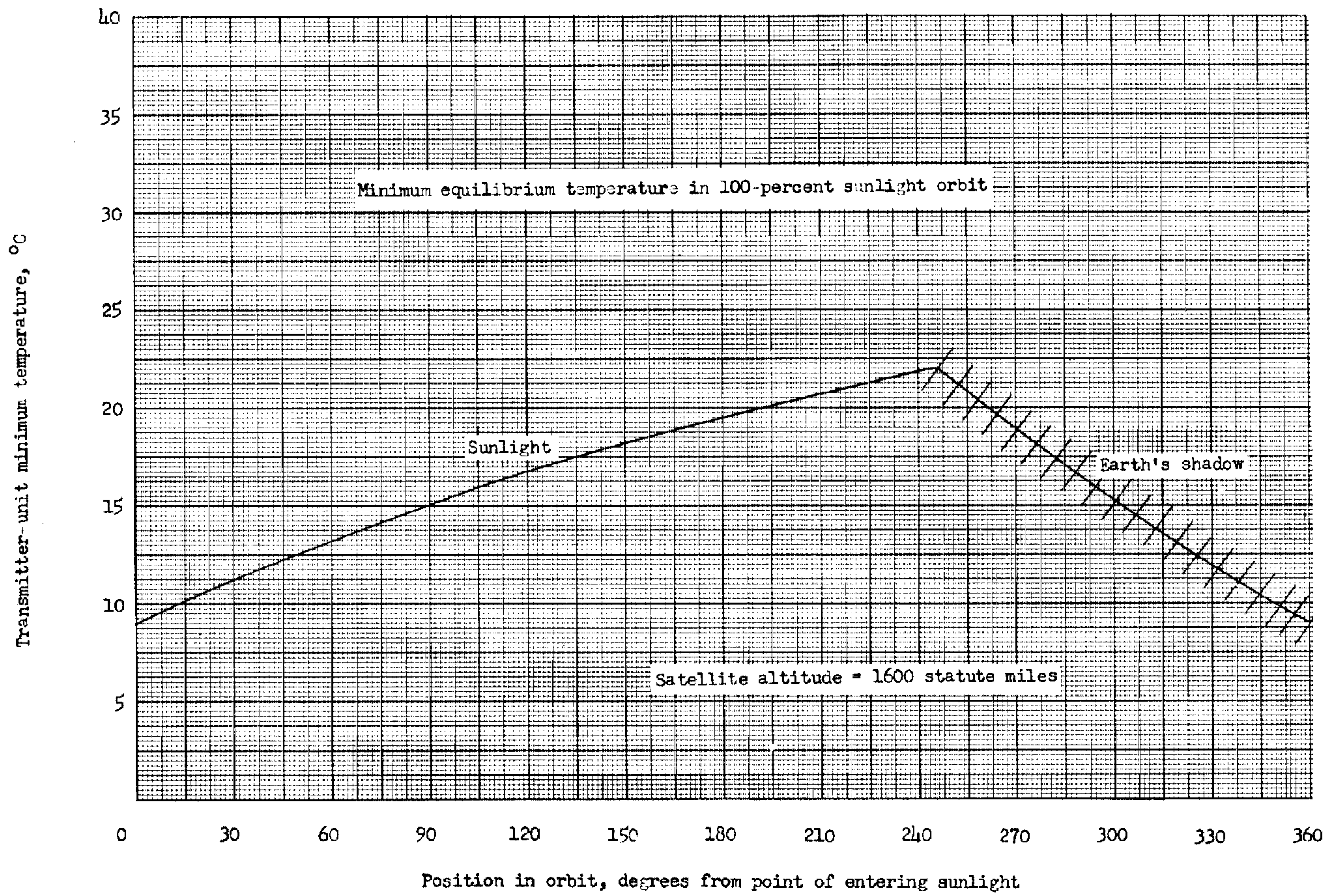


Figure 11.- Thermal response of the transmitter unit after the stabilized cycle is obtained. Orbit for the 12-foot-diameter satellite with 17 percent of the outside surface area coated with white epoxy enamel consists of 68-percent sunlight time and 32-percent shadow time.

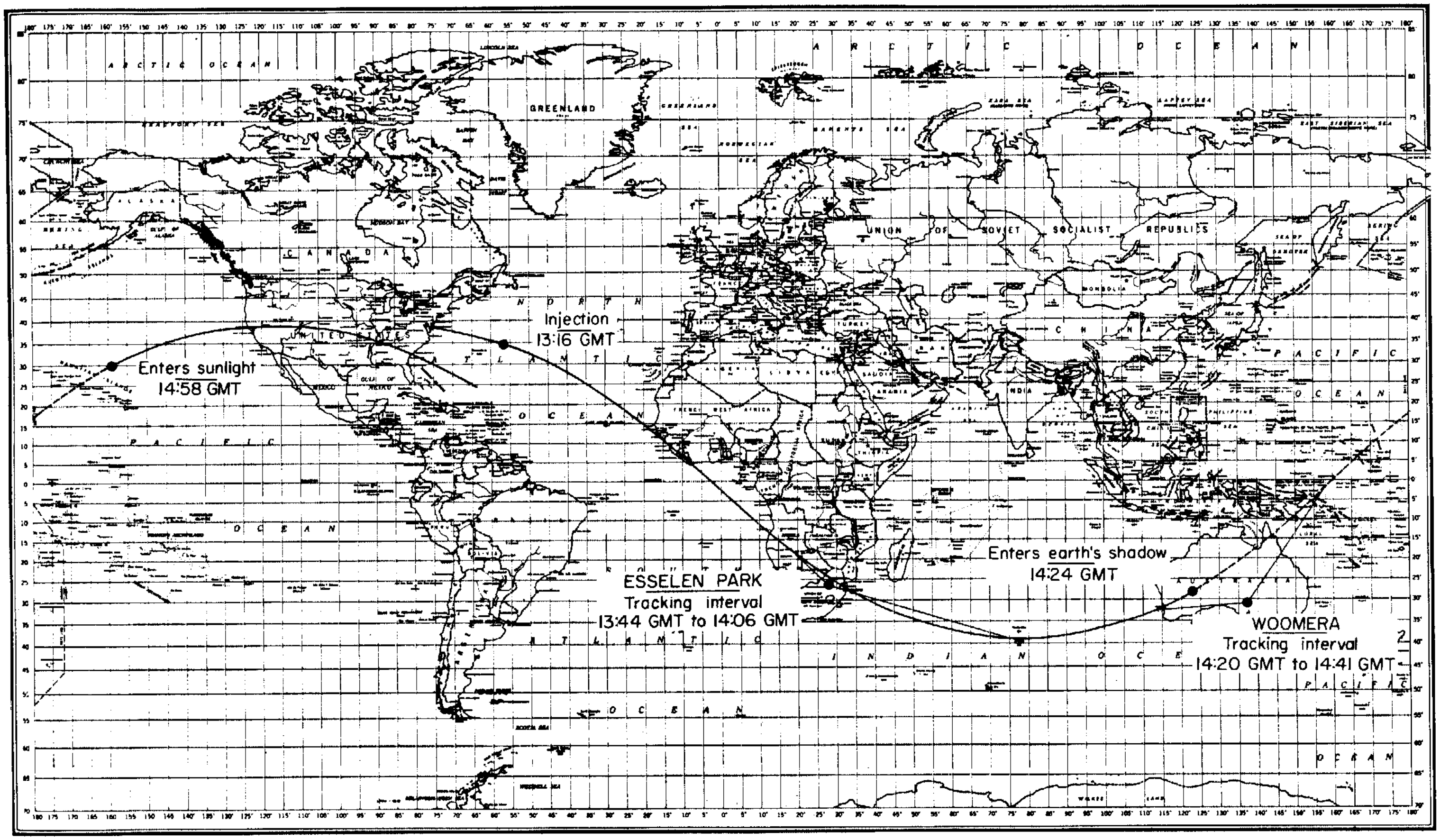


Figure 12.- Earth track for first orbit of Explorer IX satellite.

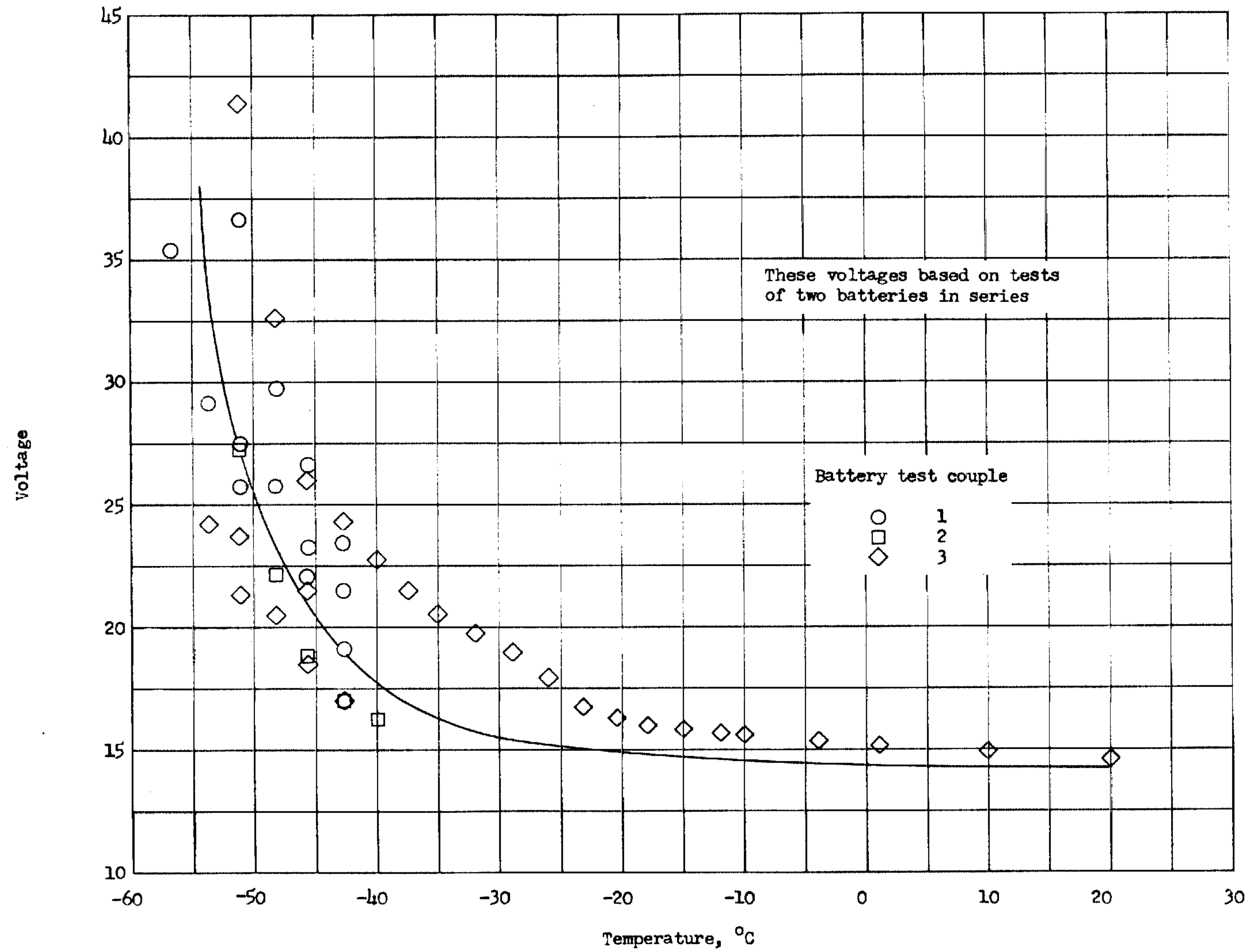


Figure 13.- Effect of battery temperature on battery charging voltage for 10-milliampere charging current.

NASA-Langley, 1962 L-1952

NASA TN D-1369
National Aeronautics and Space Administration.
TEMPERATURE CONTROL OF THE EXPLORER IX SATELLITE. Charles V. Woerner and Gerald M. Keating. July 1962. 38p. OTS price, $1.00. (NASA TECHNICAL NOTE D-1369)

A thermal analysis of the radio beacon on the Explorer IX (1961 Delta 1), an inflatable satellite having a 12-foot diameter, showed that it could be maintained within permissible temperature limits by (1) using a plastic film for the inner surface of the satellite to increase its emittance, (2) partially covering the outer surface with white epoxy enamel in order to decrease the ratio of solar absorptance to emittance, and (3) physically separating the beacon set from the satellite skin. Reasons for failure of the beacon are proposed.

Copies obtainable from NASA, Washington

I. Woerner, Charles V.
II. Keating, Gerald M.
III. NASA TN D-1369

(Initial NASA distribution: 17, Communications and sensing equipment, flight; 18, Communications and tracking installations, ground; 25, Materials, engineering; 35, Power sources, supplementary; 47, Satellites; 53, Vehicle performance.)

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NASA